

Single Stage to Orbit Mass Budgets Derived from Propellant Density and Specific Impulse*

Abstract

The trade between specific impulse and density is examined in view of SSTO requirements. Mass breakdowns for vehicle hardware are derived directly from these two key properties, for several propellant combinations. This comparative analysis, based on flight-proven launcher hardware, reveals that none of the rocket propellant combinations considered is the obvious choice for SSTO. In particular, the higher density of several alternative propellants compensates for reduced Isp, when compared with cryogenic oxygen and hydrogen. The methodology used here amounts to comparing vehicle designs, so the graphs may be refined using higher fidelity vehicle mass models. This method is therefore additionally useful for comparing the payload fractions of different rocket propelled and air breathing SSTO vehicle designs.

Introduction

The ideal chemical rocket propellant would have both high specific impulse (Isp) and high density. Unfortunately one must choose between these two desirable characteristics when selecting real propellants. Figure 1 graphically illustrates that an ideal propellant doesn't exist. However, it has been widely assumed that cryogenic oxygen and hydrogen are the best propellant pair for single-stage-to-orbit (SSTO) rockets, purely on the basis of high Isp.

Alternatively, the use of dense liquids has been advocated in order to enable extremely high propellant mass fractions. Clapp and Hunter (1993) calculated that the density of nearly-pure hydrogen peroxide (high test peroxide, HTP) and kerosene makes this propellant combination superior to oxygen and hydrogen. Their calculation assumed that the entire empty mass of a SSTO vehicle scales as propellant volume, which is true only for tanks. The need for a more accurate analysis of the trade between Isp and density was noted by these authors.

The purpose of this paper is to present a fair comparison of propellants in view of SSTO requirements. A related goal is to keep the analysis simple, so that its conclusions are independent of complex, debatable assumptions. Accordingly, approximate historical data for launcher hardware capabilities are used in this paper, instead of fundamental calculations. It is hoped that this approach will permit understanding and appreciation by a wide readership. In any event, obtaining extreme precision in a density-Isp trade is probably less important than other considerations in choosing propellants.

Simple algebraic manipulation cannot produce a scalar figure of merit to quantify the relative importance of Isp and density for high Δv missions, because the trade is specific to the mission and to actual hardware capabilities. Consider the product of density and Isp, which increases if either is individually increased, as would be expected of an appropriate figure of merit. However, this product merely represents impulse per unit volume, which would unfairly favor the dense propellants having relatively low Isp.

In this paper, idealized SSTO vehicle designs are derived from propellant density and Isp, given known hardware capabilities and the basic equations which govern rocket design and performance. The intent of the straightforward analysis is to instill "figure of merit" character in the results. In support of this goal, ratios are used instead of assuming a particular vehicle size or payload capacity. The primary intent is to fairly quantify the density-Isp trade, and not to design a vehicle.

Gross mass is obviously important for upper stages, which must be lifted by another stage when full of propellants, but gross mass is fundamentally less important for first stages and SSTO vehicles. If cost is of interest, for example, the mass of manufactured hardware is a better indicator than gross mass, since propellant is relatively inexpensive. For these reasons, Max Hunter (1989) has advocated judging SSTO vehicle designs on the basis of payload mass as a fraction of empty (hardware) mass, rather than as a fraction of gross mass (hardware + propellants). This criterion was also advocated by Sponable (1995) and is adopted here. However, note that residual propellant and pressurant are also included, as required for a consistent derivation of mass breakdowns starting with the rocket equation.

The present work calculates the contributions of engines, tanks, and residual fluids to the orbiting mass of idealized SSTO vehicles. The remaining fraction of orbiting mass represents the allowance for payload, as well as subsystems which are mostly independent of the propellant choice. Solid propellants are briefly noted but are beyond the scope of the full analysis, because major hardware differences for solid and hybrid motors increase the difficulty of a fair comparison to liquid systems.

Of fundamental importance is the required mass ratio (launch/orbiting), which is dictated by the rocket equation given I_{sp} and the required velocity increment (Δv). Higher I_{sp} reduces the required mass of propellant relative to hardware. High propellant density, which is also a key virtue, decreases tank volume, and hence the fraction of orbiting mass which must be devoted to tanks. Although the effect on engines is slightly more subtle, higher propellant density tends to reduce the required size of flow passageways, and hence the fraction of vehicle hardware mass which must be devoted to engines and related plumbing.

Propellants and vehicle mass ratios

For each propellant combination considered, the theoretical vacuum I_{sp} is used (chamber pressure = 1000 psia, area ratio = 40). A round number, 10 km/s, is taken as the Δv required to reach low earth orbit. It is assumed that the amount in excess of orbital velocity (7.8 km/s) accounts for gravity and drag losses, as well as a lower delivered I_{sp} , as limited by engine design and atmospheric operation during part of the trajectory.

Table 1 lists propellant combinations, their characteristics, and the results of applying the rocket equation. From the mass ratios, it is clear that the I_{sp} variation among the different propellants results in a wide variation in launch mass, with oxygen and hydrogen permitting the least launch mass for a given orbiting mass. The last column in Table 1, however, indicates an opposite trend for propellant volume. For a given orbiting mass, the denser propellants result in smaller vehicles, in spite of reduced specific impulse. The often-overlooked importance of propellant density can thus be appreciated at the outset. The key issue to be addressed in sections below is the fraction of orbiting mass which must be devoted to propulsion hardware (engines, tanks) and residuals for the various propellant choices.

Table 1. Characteristics of candidate propellants, with SSTO mass & volume ratios

propellant combination	O/F ratios		specific gravity			Vacuum I_{sp} (e=40)	mass ratios to achieve 10 km/s Δv			
	mass	vol	ox	fuel	bulk		launch orbiting	propellant orbiting	propellant launch	prop vol. orb mass
O ₂ —H ₂	6.00	0.37	1.14	.071	.363	452 sec	9.6	8.6	.895	23.7 l/kg
O ₂ —CH ₄	3.45	1.25	1.14	.415	.821	369	15.9	14.9	.937	18.1

O ₂ —RP-1	2.77	1.96	1.14	.810	1.03	358	17.3	16.3	.942	15.8
98% H ₂ O ₂ —JP-5	7.00	4.01	1.43	.820	1.31	327	22.7	21.7	.956	15.2
HTPB-AP-AI					1.8	310	26.9	25.9	.963	14.4

Engines

Numerous existing launch engines having thrust levels in the 1-2 MN range are of interest here because these have the highest demonstrated thrust/weight ratios. Typically, engine thrust/weight ratios at sea level are near 50 for oxygen-hydrogen propellants, and near 100 for oxygen-hydrocarbon propellants (McHugh, 1995). This significant variation results from propellant properties, since low-density propellants (particularly liquid hydrogen) require larger flow passageways and larger pumps for a given mass flow.

Examples of oxygen-hydrogen engines are the Space Shuttle Main Engine (Fisher 1995), the Vulcain engine developed for Ariane 5 (Brossel et al 1995), the J-2 engine used on the Saturn V vehicle (Fisher 1995), the LE-7 on Japan's H-II launcher (Fukushima & Imoto 1994), and the RD-0120 Energia core engine (Rachuk et al 1995). Examples of oxygen-kerosene engines include the NK-33 developed for the N-1 Russian moon vehicle (Lacefield & Sprow 1994), and the H-1 used on the Saturn IB (Fisher 1995). All examples listed here support the thrust/weight ratios indicated above and in Table 2 for oxygen-hydrogen and oxygen-kerosene engines.

Although there have been launchers which used the less-common propellants under consideration here (e.g. Parkin, 1975), there are essentially no such engines in the thrust class (and hence thrust/weight capability) discussed above. Therefore, estimated thrust/weight ratios in Table 2 are based on interpolation or extrapolation depending on propellant density.

In Table 2, the first column is obtained from Table 1. For each propellant combination, the launch/orbiting mass ratio is multiplied by a constant ratio of sea level thrust to launch weight, assumed to be 1.3. The result is then divided by the engine thrust/weight ratio, to obtain the fraction of SSTO vehicle orbiting mass which must be devoted to engines.

The last column in Table 2 reveals that the advantage of higher engine thrust/weight ratios enabled by high propellant density is almost exactly cancelled by the need to lift a greater amount of these low-Isp propellants. Clearly, a propellant choice for the SSTO mission cannot be made on the basis of proven launcher engine capabilities alone.

Table 2. Engine ratios for candidate SSTO propellants

propellant combination	<u>launch mass</u> orbiting mass	<u>sea level thrust</u> engine weight	<u>engine mass</u> vehicle orbiting mass
O ₂ —H ₂	9.6	50	0.25
O ₂ —CH ₄	15.9	90	0.23
O ₂ —RP-1	17.3	100	0.22
H ₂ O ₂ —JP-5	22.7	120	0.25

Tanks

It should be noted here that low tank pressures (<0.5 MPa) and pump-fed engines are inherent to the numbers presented in this paper. Considering basic pressure vessel equations, the ratio of propellant mass to tank mass is expected to be proportional to propellant density, if tank pressure and material properties are held constant. One might argue that larger tanks (e.g. for hydrogen) are more than proportionately heavier due to greater bending moments of longer vehicles. One might alternatively argue that a hydrogen tank is less than proportionately heavier, due to the lower elevation pressure contribution to structural loading. These subtle arguments, as well as calculations of tank performance from material properties and internal pressures, are not necessary for the purpose of this paper. The weights and propellant capacities of real launcher tanks are known, and reality accurately accounts for the relevant considerations.

Data from existing tanks are consistent, to within several percent, with estimates that could be made using the simplest pressure vessel equations. That is, tank performance is directly proportional to propellant density, with no significant dependence on absolute size. Note that associated structures such as intertanks, thrust structures, and skirts, are not considered here. Atlas tanks contain just over 100 times their own mass in oxygen-kerosene (Martin 1989), which has a bulk specific gravity just above 1 (see Table 1). Centaur tanks are essentially identical in construction to Atlas tanks, but carry only 35 times their weight in oxygen-hydrogen (Richards 1989).

Historically, aluminum tanks were less efficient than the Atlas-Centaur stainless steel tanks (Midgley 1970). For example, the Saturn V first stage oxygen tank carried only 81 times its weight in liquid oxygen (Boeing 1968), which has a specific gravity of 1.14. However, the Shuttle oxygen tank now carries 115 times its weight in oxygen (Lockheed-Martin 1995). The ratio is only 8 for the Shuttle hydrogen tank, which is consistent with hydrogen's specific gravity of .07. Finally, the combined propellant/tankage mass ratio for the Shuttle is 39, very close to that of the much smaller Centaur. Note that the masses of the Intertank and other subsystems of the Shuttle External Tank are omitted here.

It can thus be appreciated that "1% tankage" exists for water density fluids, and that the actual mass ratio of propellant to bare tanks varies linearly with propellant density. Therefore, the ratios of propellant mass to tank mass listed in Table 3 are simply 100 times the bulk specific gravity, from Table 1. The first column in Table 3 is borrowed from Table 1. Dividing the first two columns in Table 3 yields the fraction of each SSTO vehicle's orbiting mass which must be tankage.

The last column in Table 3 has profound implications. Based on historical tank capabilities, more than one quarter of the all-cryogenic SSTO orbiting mass must be tankage. However, SSTO vehicles designed for kerosene fuel need only devote one-sixth of their orbiting mass to tankage.

Table 3. Tank mass ratios for candidate propellants

<u>propellant combination</u>	<u>propellant mass orbiting mass</u>	<u>propellant mass tank mass</u>	<u>tank mass orbiting mass</u>
O ₂ —H ₂	8.6	36	0.26
O ₂ —CH ₄	14.9	82	0.18
O ₂ —RP-1	16.3	103	0.16
H ₂ O ₂ —JP-5	21.7	131	0.16

Residual fluids

This section presents estimated contributions of residual propellant and pressurant to the SSTO orbiting masses, for the various propellants.

Table 4. Residual fluid mass ratios for candidate SSTO propellants

<u>propellant combination</u>	<u>liquid residual total liquid</u>	<u>liquid residual vehicle orbiting</u>	<u>ullage residual total liquid</u>	<u>ullage residual vehicle orbiting</u>
O ₂ —H ₂				
O ₂ —CH ₄				
O ₂ —RP-1				
H ₂ O ₂ —JP-5				

Final Results

This section sums the various contributions to the orbiting masses, and graphically displays the results. The significance of the mass fractions which remain unaccounted for is discussed in terms of allowance for payload and other subsystems.

Other characteristics of candidate propellants

This section briefly discusses additional factors which would affect propellant choice, such as ease of operations, safety, insulation required, etc.

Discussion

This section provides a conclusion, and discusses the author's opinions about SSTO vehicle design. Tripropellant engines, and the implications of advancing structural technology, will most likely be touched upon.

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